THE INTEGRATION, TESTING AND FLIGHT OF THE EO-1 GPS

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ABSTRACT

The Global Positioning System has long been hailed as the wave of the future for autonomous on-board navigation of low Earth orbiting spacecraft despite the fact that relatively few spacecraft have actually employed it for this purpose. While several missions operated out of the Goddard Space Flight Center have flown GPS receivers on board, the New Millenium Program (NMP) Earth Orbiting-1 (EO-1) spacecraft is the first to employ GPS for active, autonomous on-board navigation. Since EO-1 was designed to employ GPS as its primary source of the navigation ephemeris, special care had to be taken during the integration phase of spacecraft construction to assure proper performance. This paper is a discussion of that process: a brief overview of how the GPS works, how it fits into the design of the EO-1 Attitude Control System (ACS), the steps taken to integrate the system into the EO-1 spacecraft, the ultimate on-orbit performance during launch and early operations of the EO-1 mission and the performance of the on-board GPS ephemeris versus the ground based ephemeris. Conclusions will include a discussion of the lessons learned.

INTRODUCTION

The Global Positioning System (GPS) is not a new idea. Neither is using the navigation capability of GPS for Low Earth Orbiting (LEO) spacecraft a new idea. Although the idea has been around for quite some time and many papers have been written regarding the uses of GPS on orbit, only a handful of spacecraft have actually used GPS receivers on-board LEO spacecraft for primary real time navigation control. After more than a decade of experiments, the Earth Observing –1 (EO-1) spacecraft has become the first satellite designed and built at NASA's Goddard Space Flight Center to baseline GPS as the primary means of performing on-board real time navigation control.

GPS On-Orbit History

The first GPS receivers to fly on orbit were aboard NASAs Earth resources satellites Landsat-4 (launched in July of 1982) and Landsat-5 (launched in March of 1984). While both of these satellites possessed the capability to use GPS data as their primary source of ephemeris state information, at that time (the early 1980's) less than half of the GPS constellation was flying, so reliable navigation fixes were simply not possible over the prolonged times required by the Landsat mission. For these two spacecraft, uploaded ephemeris files were the main means of navigation and so GPS amounted to little more than an experimental footnote. Ten years later and although the GPS constellation was complete it was still not being used for active on-board navigation and attitude reference frame computations. Since that time, a host of spacecraft bearing GPS receivers have followed, most of which included GPS for experimental purposes only. **Table 1** [Ref 1] shows an extensive but by no means exhaustive list of such spacecraft.

Launch Target	Spacecraft	GPS Receiver	GPS Function
7-1982	Landsat-4	Motorola GPSPAC	Constellation incomplete - Experimental
3-1984	Landsat-5	Motorola GPSPAC	Constellation incomplete - Experimental
7-1991	ORBCOMM-X	--	Launch Failure
6-1992	EUVE	Motorola GPSDR	GPS Orbit Determination Experiment
8-1992	TOPEX/Poseidon	Motorola GPSDR	GPS Orbit Determination Experiment
multiple	Space Shuttle	multiple	Nav, Relative Nav & Att Experiments
6-1993	RADCAL	TANS Quadrex	Post Flight Attitude Experiment

7-1993	ORFEUS-SPAS-1	Alcatel/SEL	
9-1993	PoSat-1	TANS	Orbit Determination
2-1994	OREX	GPSDR	
3-1994	DARPASAT	AST-V	Orbit Determination
5-1994	TAOS/STEP-0	AST-V	Orbit Determination
5-1994	STEP-2	AST-V	Orbit Determination
11-1994	CRISTA-SPAS	TANS Vector	Attitude & Orbit Determination
'92 & '94	COMET	Ashtech SB24	
1-1995	Faisat-1	--	
3-1995	SFU	GPSR	
4-1995	ORBCOMM-FM1	TANS II	
4-1995	ORBCOMM-FM2	TANS II	
4-1995	OrbView-1	TurboStar	
9-1995	Wake Shield Facility-02	TurboStar & Tensor	Attitude & Orbit Determination
1-1996	GADACS / SPARTAN	Two TANS Vectors	Receiver Software Error
5-1996	GANE / STS-77	TANS Vector	Real-time Nav & Att Experiments
5-1996	MSTI-3	Viceroy	
5-1996	MOMS-2P	Viceroy	
11-1996	HETE	SEXTANT	
11-1996	Wake Shield Facility-03	multiple	Navigation Experiment
11-1996	ORFEUS-SPAS	Tensor	Relative Nav Experiment
2-1997	HALCA	GPS	
3-1997	Zeya	GPS and GLONASS	
8-1997	OrbView-2	redundant Viceroys	
8-1997	SSTI Lewis	Two Tensors	S/C Failed shortly after launch
9-1997	Faisat-2v	--	
9-1997	IRS-1D	--	
10-1997	Falcon Gold	TIDGET	
10-1997	YES	TANS II	Orbit Determination above GPS Constellation
11-1997	ETS-VII	--	Rendezvous and Docking Experiment
12-1997	Equator-S	Viceroy	Orbit Determination above GPS Constellation
12-1997	EarlyBird	Vector and Viceroy	
2-1998	GFO	Four TurboStars	Precise Orbit Determination
2-1998	Globalstar	Tensor	Orbit Determination and Time
2-1998	SNOE	MicroGPS (Bitgrabber)	
7-1998	FASat-Bravo	TANS II	
7-1998	TMSat-1	SGR-10	
10-1998	SEDSat-1	Ashtech G12	
10-1998	ARD	--	
11-1998	International Space Station	SIGI	Operational Navigation Attitude & Time
12-1998	SAC-A	TANS Vector	Orbit Determination & Attitude Experiment
1-1999	ORSTED	TANS, TurboStar	
2-1999	ARGOS	--	
2-1999	SUNSAT	TurboStar	Orbit Determination
4-1999	UoSAT-12	SGR-20	Orbit Determination
4-1999	Ikonos-1	Rockwell C/A code	
5-1999	IRS-P4 (OceanSat)	--	
6-1999	QuikSCAT	2 Viceroys	
9-1999	SRTM	Blackjack	
9-1999	JAWSAT	TANS Vector	
late 1999	AMSAT Phase 3D	Two TANS Vectors	Orbit & Attitude Det. above GPS Constellation
1999	STRV-C	Blackjack	
1998	ARISTOTELES	--	
1997	EOS-A	--	
1998	EOS-B	--	
1998	TSX-5	two TANS Vectors	Orbit Determination
1999	SAC-C	Lagrange, Tensor, Bitgrabber	
1999	QuickBird	2 Viceroys	
(late '90s)	European Polar Platform	--	
(late '90s)	RAMOS	--	
(1999)	Gravity Probe B	2 Vectors	Has yet to launch
(12-1999)	CHAMP	Blackjack	
(1999)	OSEM	Tensor or TANS Vector	

(5-2000)	Jason-1	Blackjack	Has yet to launch (summer 2001?)
(Aug 2000)	VCL	Blackjack	Has yet to launch (2002 ?)
(2000)	BIRD	Rockwell Collins	

Table 1: An incomplete history of GPS receivers in space.

...and there are plenty more to follow. In 1995 NASA Management Instructions directed that the lowest cost navigation system should be implemented on all NASA spacecraft. Since GPS was seen as a low cost alternative to standard practices, this instruction (which has since been superceded by the NASA Operational Directive Information System) was promptly interpreted in some quarters as to mean that all LEO spacecraft should include GPS receivers. Such was the origin of a GPS capability on NASA the New Millenium Program Earth Observing spacecraft EO-1. This spacecraft was to be a testbed for demonstrating new concepts in autonomy and advanced technologies and thus was a natural choice for showcasing GPS capabilities on orbit.

EO-1 ACS DESIGN

The EO-1 Attitude Control Subsystem establishes and maintains the pointing of the EO-1 spacecraft science instruments at desired targets while allowing for the proper orientation and operation of other spacecraft subsystems such as power, thermal and communications. Attitude control functions are performed by a suite of on-board attitude sensors, control actuators and mathematical control laws residing in a Mongoose 5 (M5) main spacecraft computer. Safe Hold Mode (SHM) controllers and electrical interfaces to most components is provided by the Attitude Control Electronics (ACE).

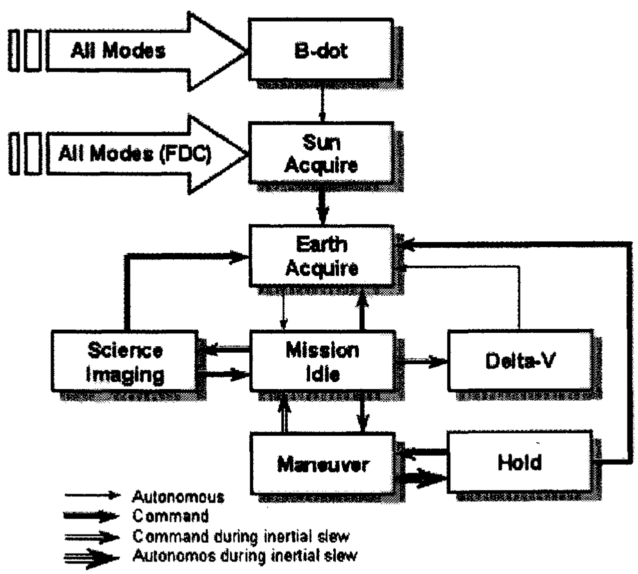

Figure 1: EO-1 ACS Mode Transitions

The EO-1 ACS flight software was derived from the GSFC Tropical Rainfall Measurement Mission (TRMM) design. For EO-1, the main ACS software on the M5 processor was to have attitude determination and closed-loop control modes for: magnetic de-spin following separation from the Delta launch vehicle, initial stabilization and sun acquisition, nadir-pointed science data collection and downlink, thruster maneuvers (for delta-V), and sun/moon slew/scan maneuvers for instrument calibrations.

The ACS flight software mode transitions are illustrated in **Figure 1**. Following separation from the launch vehicle, the ACS was to null the tip-off rates via a B-dot magnetic control law and stabilize the spacecraft before, during and after solar array deployment. During initial sun acquisition and "safe-hold" operations, the spacecraft would maintain an inertially fixed, solar-pointing attitude with the instruments away from the sun. During normal operations, the body-fixed science instruments point toward the earth with the center of their FOVs along the nadir axis. During sun and moon calibrations, the instruments point toward those bodies.

During launch, the GPS receiver would be powered off. A predicted ephemeris would be uploaded prior to launch as part of the pre-launch Absolute Time Sequence (ATS). This pre-launch orbit initialization was not a requirement for launch since orbit knowledge is not required for either B-dot or Sun Acquire modes. The initial predicted orbit would require further updating once the spacecraft was on orbit to account for any deviation in the launch vehicle trajectory. Shortly after launch and separation, the GPS would be powered on, configured and initialized for orbital operations. Until the GPS could be determined to be functioning on orbit, ephemeris uploads from the ground would be the method of choice for computing navigation and attitude reference solutions. Once outputs from the GPS were proven to be accurate, its navigation solutions would replace the ground loaded ephemeris in the active control loop.

GPS AS PART OF EO-1 ACS

Part of the charter of the New Millenium Program (NMP), of which EO-1 is a part, is to demonstrate new technologies that can be used to improve performance or lower the cost of future spacecraft . While GPS is hardly a new technology, until EO-1, no spacecraft operated by the Goddard Space Flight Center had ever been baselined to employ an on-board GPS receiver as the primary method of ephemeris estimation. The on board ephemeris estimation would be used to determine the attitude reference frame against which attitude measurements are compared. EO-1 would be the first spacecraft to employ GPS in this operational capacity. The space-qualified Space Systems/Loral Tensor™ was the unit selected to fly on EO-1. It is capable of providing 1 Hz position and velocity data in Earth Centered Inertial (ECI-J2000) coordinates, along with an absolute time reference. With internal Kalman filter software enhancements, the 3-sigma accuracies for position and velocity are ±150 meters and ±0.55 meter/second respectively. The one Pulse Per Second (PPS) coordinated universal time (UTC) reference output is accurate to ±1 microsecond.

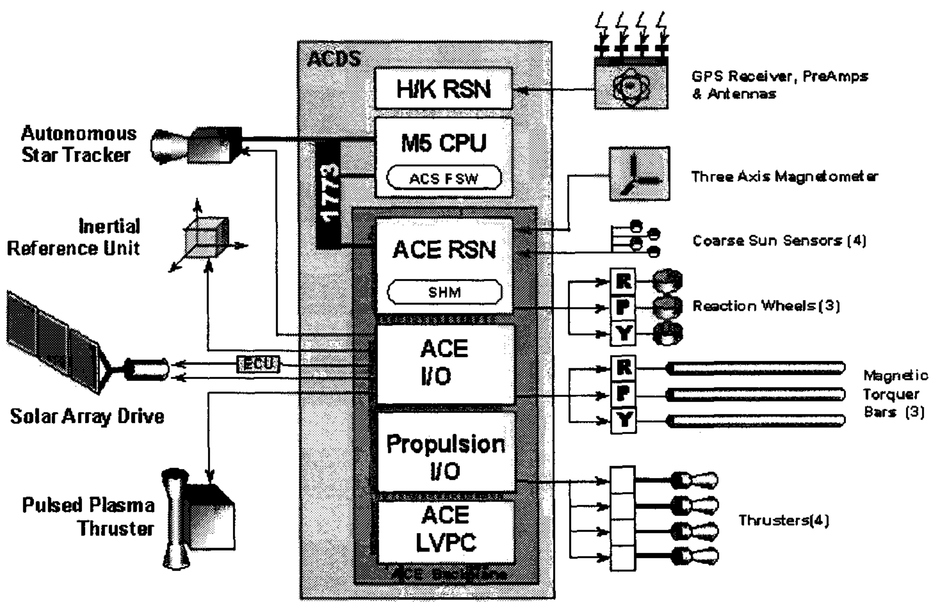

Figure 2: EO-1 Attitude Control System Components

The EO-1 design called for GPS data to be transferred via serial digital link to the Housekeeping RSN, and then read by the main ACS M5 processor over the 1773 bus shown in **Figure 2** Because the Tensor™ receiver was designed for a different power bus, a separate Power Conditioning Unit (PCU) was needed to provide the receiver 29±3 VDC from the 28±7 VDC EO-1main power bus. The components of the EO-1 GPS are shown in Figure 3.

The Tensor™ is a nine channel dual receiver. That is to say, the off the shelf unit consists of two redundant GPS receivers each of which can accumulate and process data from up to nine GPS SVs simultaneously. EO-1 required only one side of this receiver (as shown in Figure 3) so that the Tensor™ (as delivered by Space Systems/Loral) was split into two single string units.

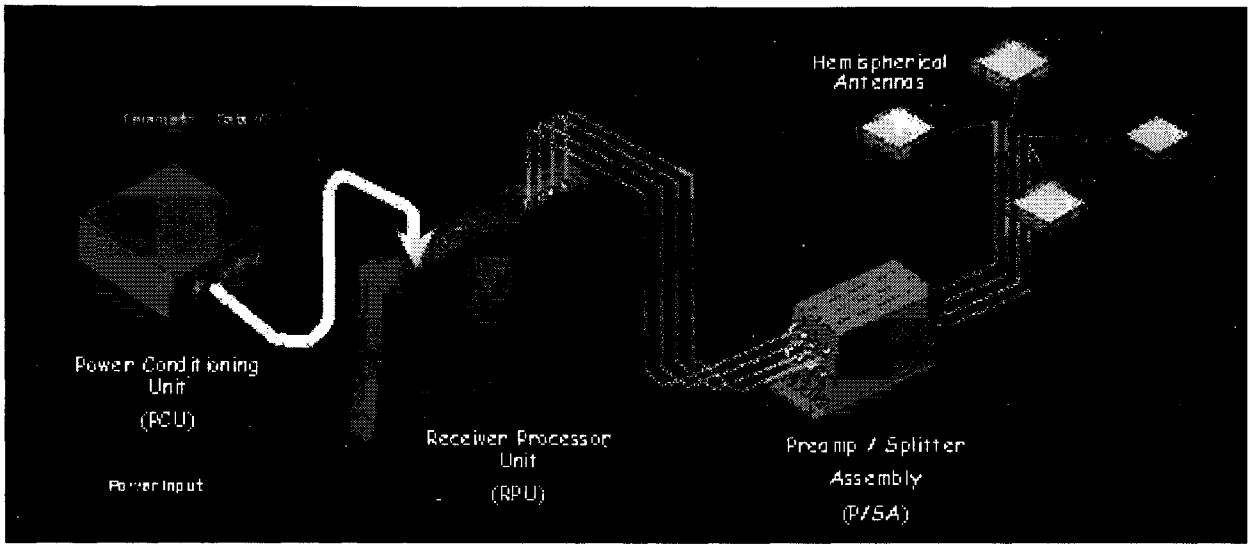

Figure 3: EO-1 Global Positioning System Components

When in operation, the GPS is primarily solving for x-y-z position and time. This is why at least four GPS Space Vehicles (SVs) must be in view providing four independent distance measurements to the receiver before an navigation solution can be attempted. If four SVs are in view, the receiver can determine the distance between the user spacecraft and each of the four GPS SVs at the broadcast time (all GPS SVs broadcast the current time). Since the locations of the GPS SVs are extremely well known, the user's position can be determined to a high degree of accuracy. This position and time data are then employed to resolve the orbital velocity. Since it is a derived quantity, velocity tends to be somewhat noisier than either time or position estimates.

Before any GPS receiver can begin computing navigation solutions, it must know three things. First it must know what time it is, it must know where it is and it must know where all the GPS SVs are as well. Only then can it determine which GPS SVs are visible and of those which are the best to use. The current time is simply that: the GPS time which is specified to be within ±1 nanosecond of UTC proper and is continually broadcast by all the SVs in the GPS constellation – if the receiver can acquire signals from even one GPS SV, it can get the time. The location of all the GPS SVs in orbit is contained in the GPS Almanac which specifies the orbital elements of all (up to 32) of the GPS SVs at a known epoch. This data is also simultaneously broadcast by all SVs in the GPS constellation and repeats every 12.5 minutes. If the receiver can lock up on even one SV for that long, it can acquire the position of all the other SVs in the constellation. Almanac files are also archived on the world wide web by the U.S. Coast Guard. The user location data is referred as the User Ephemeris data. Providing the receiver initial estimates of Time, Almanac and Ephemeris data allows the receiver to begin computing solutions within five minutes and is referred to as a 'warm start'. Waiting for the receiver to collect the time and Almanac data and then find four SVs on its own to begin computing navigation solutions is called a 'cold start' and can take considerably longer to accomplish.

There are two switches that must be configured before GPS outputs would be permitted to control the EO-1 spacecraft on orbit. The first enables ACS processing of on-board GPS ephemeris data so it can be monitored in telemetry, but would not enable the use of the GPS solution for orbit determination. The orbit solution propagated from the uploaded ground ephemeris data would continue to be used for on board orbit determination. The second switch would enable the use of the GPS solution for orbit determination so long as the solution is deemed valid. A GPS solution is considered valid if the GPS data packet containing the solution is valid, if the GPS packet checksum is good and if the health status is good and has been for at least five cycles. If all these conditions are met and the GPS propagated data passes the continuity checks, then and only then could the GPS solution be used for active on-board orbit determination.

When GPS is being used for on-board orbit determination, it provides ephemeris updates at 1 Hz. Each GPS data packet must pass all validity checks and the ephemeris data, propagated to the current cycle time, must pass the

same continuity checks described above for the ephemeris upload command. In this case, the position continuity tolerance is set to 50 km and the velocity continuity tolerance to 0.5 km/sec. If the GPS packet is invalid, the GPS packet checksum is bad, the GPS health status word is bad or the GPS propagated data does not pass the continuity checks, a GPS failure counter is incremented. The last valid ground based orbit solution is always propagated to the current time, and this solution is used instead of the GPS solution when the GPS solution fails at least one of the validity checks. The next valid GPS solution will clear the failure counter to zero, but if the GPS failure counter reaches the limit of 60 counts (this was later changed to 120 counts), GPS processing would be disabled.

EO-1 GPS INTEGRATION & TEST

Since the December 31, 1997 delivery of the GPS flight hardware from Space Systems / Loral to the GSFC Guidance Navigation & Control Center (and the subsequent acceptance testing performed at Goddard by Space Systems / Loral prior to formal acceptance), the unit successfully completed an extensive battery of tests designed to validate system compliance with design specifications and mission requirements as well as to characterize system performance under various circumstances which were expected to be encountered by the EO-1 spacecraft. This testing was accomplished in five phases over a period of several months and executed in order of importance to the project subject to resource availability

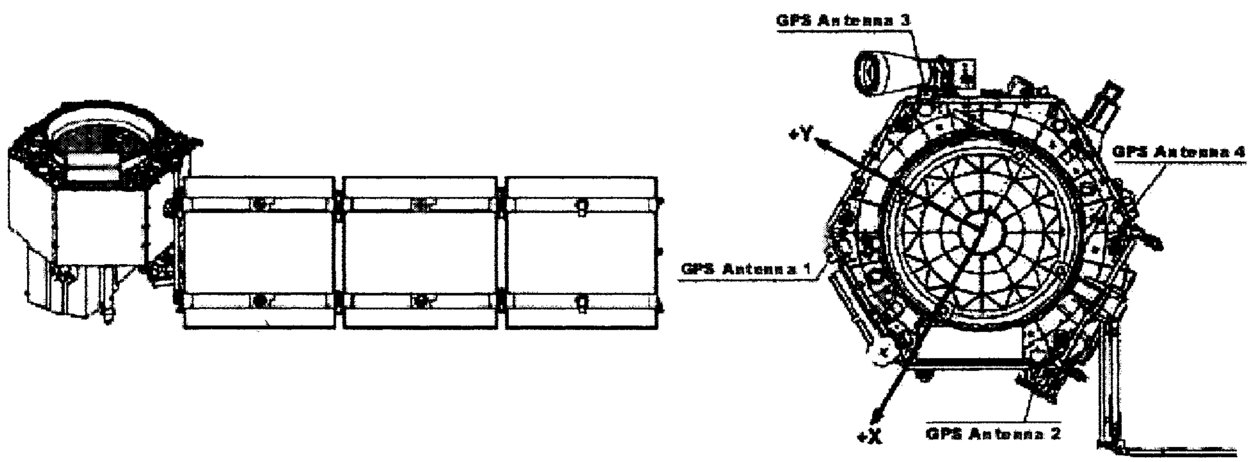

Figure 4: EO-1 Spacecraft and the Locations of the GPS Antennas on the Zenith Deck.

Phase I: Requirements Compliance (Performance) Testing:

The first sequence of tests on the EO-1 GPS Flight Receiver employed the GPS Simulator. For these tests, a single 6-hour simulator scenario was assembled which modeled the expected EO-1 orbital characteristics:

SemiMajor Axis:	7083.0 km	Longitude of A.N:	61.2 deg
Eccentricty:	0.00054	Argument of Perigee:	-37.9 deg (322.1 deg)
Inclination:	98.2 deg	Mean Anomaly:	0.00 deg

Epoch: 20-Mar-1998 00:00:00 UTC

The spacecraft characteristics were modeled to conform with those expected of the EO-1 spacecraft, the most significant of which is the non-nominal antenna pointing directions. Unlike standard GPS antenna orientations, where all antennas point in the expected zenith direction, the EO-1 GPS antennas were all canted 45° away from zenith so as to avoid interference and obscuration by the payload attach fitting (PAF) also mounted to the zenith deck of the spacecraft a seen in **Figure 4**. The PAF was present to attach EO-1 to the Delta launch vehicle. To model the EO-1 GPS antennas, the following locations and orientations for the antenna centers were observed:

	x-center	y-center	z-center	Azimuth	Elevation
Antenna-1:	0.3955 m	0.5927 m	-0.5040 m	30 deg	-45 deg
Antenna-2:	0.3955 m	0.5173 m	-0.5040 m	330 deg	-45 deg
Antenna-3:	-0.4705 m	0.5927 m	-0.5040 m	150 deg	-45 deg
Antenna-4:	-0.4705 m	-0.5173 m	-0.5040 m	210 deg	-45 deg

The x-y-z centers locate the center of the antenna in the spacecraft navigation frame (origin at the spacecraft center of mass) and where azimuth is measured in the navigation x-y plane with the x-axis as a zero reference and is positive towards the y-axis (counter-clockwise about the +z-axis). Elevation is measured off the x-y plane and is positive towards the positive z-axis.

A zero attitude error Earth pointing simulation scenario beginning at the design epoch and lasting for 6-hours was constructed. All subsequent Earth pointing tests in Phase I were executed using this scenario. For each test of the flight receivers (both sides were tested in parallel even though only one side would ultimately fly), they were powered up and placed in the same 'test specific' configuration. Almanac data was then provided to the receivers via laptop computer, along with initial estimations of ephemeris and time, thus allowing the unit to execute what will be henceforth referred to as a 'warm start'. This provides the receiver with enough information to attain an immediate (< 5mins) lock on the GPS constellation and begin providing navigation solutions, thus streamlining the acquisition process by avoiding the time consuming 'cold start' process which starts the receiver from an unknown navigation state and requires the receiver to initiate a random search for GPS satellites. The point of this first series of tests was performance evaluation, not cold start characterization, which was to be the subject of later testing.

Configuration Test 1: Default

For the first test, the default mode was tested. In the default mode, the receiver is free to switch back and forth between the Single Point Solution (SPS) and the embedded Kalman Filter outputs. This configuration assumes that the attitude function of the receiver should be switched on, even though attitude outputs were not to be employed by EO-1. In fact, since the attitude portion of the receiver software assumes that all antennas are pointed directly toward zenith, the canted configuration of the EO-1 antennas renders the attitude computations of the receiver virtually useless.

Configuration Test 2: Attitude Off

For the second test, attitude computations were switched off, demonstrating that there was no difference in navigation performance of the receiver whether the attitude computations are left on or switched off.

Configuration Test 3: Force SPS

For the third run, the embedded Kalman filter was not employed, yielding SPS data only for navigation. While in this mode, the receiver collects data from the GPS constellation and computes one navigation solution at a time (thus the name Single Point Solution). Each solution is independent of the last so solutions can be generated in this mode only when four or more GPS Space Vehicles (GPS-SVs) are visible. During those occasional periods when geometry is not sufficient to compute a single point solution, no data is output and the receiver is said to experience a 'navigation outage'. This test revealed outages to be a fairly frequent occurrence averaging five or more brief outages per orbit.

Configuration Test 4: Force Filter Solutions

For the last run, solutions from the filter was were output exclusively. As might be expected performance with the filter is considerably better than exclusive use of SPS data. Not only does the filter smooth the SPS data reducing overall errors, but the memory inherent in a Kalman filter allows the receiver to 'flywheel' through those outage periods were SPS solutions are unavailable. Only when SPS solutions are unavailable for an extended period of time does the filter show any signs of diverging. This can be seen by comparing the SPS plots (which show the SPS navigation outages) with the filtered output.

Conclusion of Phase I:

This first series of tests demonstrated that the Loral Tensor™ met the EO-1 navigation requirement and would meet the mission objectives [Ref 3].

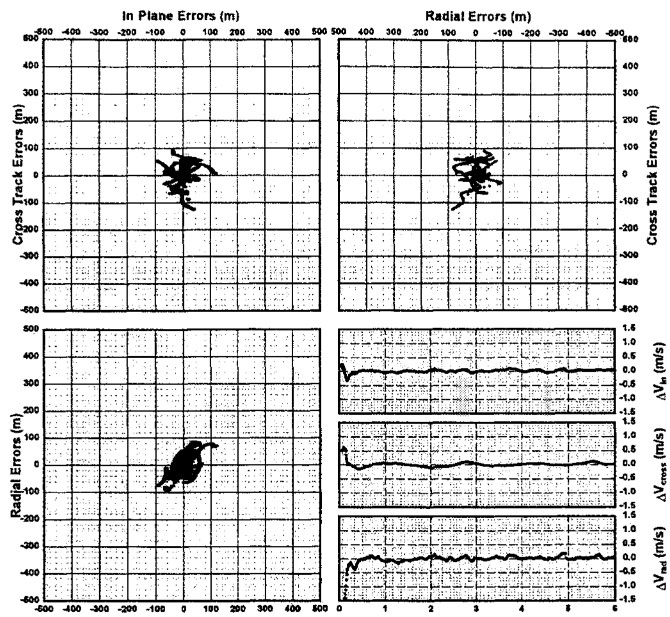

Figure 5: 3D Filtered Position & Velocity Errors from the Phase I simulation study.

	In-Track	Cross-Track	Radial	
Position:	100	130	100	m, 3σ
Velocity:	0.15	0.25	0.25	m/s, 3σ
Stability:	15.0	3.0	5.0	m over 1 sec intervals, 3σ

Table 2: Tabulated Results from the Phase I simulation study.

Phase II: Integration & Testing Plan Development:

The second sequence of tests validated the Ground Support Equipment (GSE) and procedures to be employed during Integration and Test (I&T) process. Each of these tests were first run in the lab in an effort to assure that all necessary procedures had been correctly developed, that all GSE hardware was available and functional, and to pre-quantify what to expect during the actual I&T process.

Test 1: Component Throughput / Aliveness:

The purpose of this set of tests was to develop an ability to verify that proper Radio Frequency (RF) power levels reach the receiver. Replicating the conditions expected during the real I&T process required a GPS antenna on the roof of the building (with a clear line of sight to the real GPS constellation) with enough RF cable to allow this rooftop antenna to act as an RF source for the flight receiver once mounted to the spacecraft (in the I&T test facility). This was expected to require no more than 120 feet of RF cable. Compensating for losses in such a long cable (which is not representative of the flight configuration) required the inclusion of a pair of powered RF amplifiers to boost the signal. The resulting GSE setup employed a rooftop GPS antenna followed by a 3 meter run of RG-142 coaxial cable. This led to a pair of powered amplifiers whose output was sent to the test chamber via approximately 200 ft of Belden 9913 low noise cable. The end of this cable was then connected to the spacecraft Preamp / Splitter Assembly (P/SA), providing signal to the receiver. This setup would allow the testing of P/SA and RPU and flight cable throughput, bypassing only the flight antennas.

Test 2: System Throughput / Aliveness:

The purpose of this test set was to develop an ability to verify proper RF power levels reach the receiver when connected to the P/SA via the flight antennas. Replicating the conditions expected during the real I&T test required the same GSE set-up as the first test, with one important addition. Here again, RF signals are captured via a rooftop antenna, amplified, passed through the long GSE cable and ultimately fed into the flight system. In this case, the flight antennas are not bypassed, but instead the signal is fed into a non-flight source antenna which re-radiates RF signal into one of the four flight antennas. The non flight source antenna is held steady about 1 inch from the flight antennas by a mechanical fixture hereafter referred to as a 'hat coupler' (four of which were constructed, one for each of the four flight antennas). The flight antennas are then connected to the Preamp via the flight cables. In this way, throughput of the complete flight system can be verified against the real GPS constellation.

The test was virtually identical to the one above, with the exception that the physical set-up was as described above, and the GSE hardware had already been constructed and its performance quantified as part of the first tests. The first objective of these tests was to demonstrate that the antenna-to-antenna RF 're-radiation' concept provided sufficient RF input to the GPS receiver for it to properly perform its navigation function. The second objective was to establish any sensitivity difference between this method of input to the GPS receiver and the method employed by the first set of tests (which by-passed the antennas). The third (and primary) objective was to verify proper RF throughput of the complete EO-1 Flight GPS, that is through all four antennas. All of these objectives were successfully accomplished.

Test 3: System Functional Performance:

As a final check, orbital performance was tested using the GPS RF simulator employing the same scenario developed for the Phase-I study. For this performance test, the simulator output is to be connected to the four GSE hat couplers, and the generated signal re-radiated into the four flight antennas in a manner identical to that employed during the previous test. Here, the configuration was the same as before, only now the simulator provided RF signals to all four antennas simultaneously. Based on this work, it was determined that the I&T procedure previously developed could be used to warm start the receiver for performance verification of EO-1 GPS during I&T.

Phase III: Timing Performance Testing:

The third sequence of tests on the EO-1 GPS Flight Receiver involved testing the synchronization Pulse Per Second (PPS) output of the receiver. Specification documentation [Ref 2] from Loral states the following:

"An RS-422 pulse shall be output within ±1.0 μsec of every GPS second"

Later, the same document states:

> " The PPS leading edge occurs sometime in the 999th millisecond of an integer clock time. The countdown process to generate the leading edge is performed in that last millisecond. The Pulse Per Second, PPS, signal is output once per second with an accuracy within one microsecond with respect to GPS time."

Which means there are two quantities which needed to be verified, one being the offset of the receiver PPS from the GPS-SV time (which is itself specified to be within 100 nanoseconds of the theoretical 'absolute' UTC) and the time difference between consecutive PPS pulses. Not all of this could be tested with the equipment residing in the GSFC GPS facility at the time. What was tested was the time difference between the trailing edge of a plus PPS output and the leading edge of the corresponding minus PPS output which was measured for every other pulse pair. In the past, taking measurements exclusively off of either plus or minus PPS outputs allowed measurements to occur only for every forth pulse, so this approach is seen as an improvement over previously employed techniques. Since the same physical oscillator drives all four PPS outputs of a given receiver (plus-1, minus-1, plus-2, minus-2), one six hour test was executed on both the A and B receivers (both of which were warm-started per the procedure developed during Phase II).

Timing tests of the two sides of the GPS Tensor™ receiver were completed to the level possible with the test equipment present in the GPS facility. The characteristics of the PPS alignment to absolute time was not measurable

in the GPS Test facility, while the PPS period was. The characteristics of the PPS period for the A and B sides of the Tensor™ were found to be as follows:

	Pulse Width	Period	Mean Error (μ)	Stand. Dev (σ)
Side-A:	0.978 μsec	1 sec $\pm$ 1 msec	0.01423 μsec	0.3791 μsec
Side-B:	0.978 μsec	1 sec $\pm$ 1 msec	0.01416 μsec	0.4152 μsec

Table 3: Tabulated Results from the Phase III timing tests.

<u>Phase IV: Other Simulation studies:</u>

While no requirements were levied on the performance of the GPS receiver during non-nominal operational modes, it was considered wise to characterize the performance during any such modes that could be expected to be encountered during the run of the EO-1 mission.

Characterization Test 1 - Sunpoint

For this simulation study, the same scenario from Phase I was employed. In it, the theoretical spacecraft was held inertialy fixed, (attitude such that the Sun remained at a fixed position in the EO-1 body frame). This was accomplished by starting the spacecraft at the Earth Pointing attitude at the beginning of the six hour run (using a warm start to quickly lock up on a navigation solution) only this time commanding the simulator to hold the Sun fixed in the body frame representing a transition from Earth Point to Sun Point. The results of this run demonstrated that the receiver in the nominal EO-1 flight configuration should be able to maintain navigation solutions with the following general characteristics:

	In-Track	Cross Track	Radial	
Position	320	120	250	m, 3σ
Velocity:	0.35	0.20	0.40	m/s, 3σ

Table 4: Tabulated Results from the Phase IV characterization tests.

Again, since there were no requirements levied on the receiver performance while in the Sun Pointing safehold attitude, these numbers were provided for the purpose of characterization only.

Characterization Test 2 - J2000 Confirmation

For this study, the theoretical spacecraft was run in the same Earth pointing mode as it was during the performance simulation studies. The key difference here being that both sides of the receiver were configured to output the navigation data in the ECI-J2000 (Earth Centered Inertial) coordinate frame rather than the default ECI-True-of-Date frame. The results of this test demonstrated that the receiver performance while configured to output navigation solutions in the J2000 coordinate frame was comparable to the True-of-Date solutions cited earlier to validate performance. This indicated that the proper rotations from Earth Centered Earth Fixed (ECEF) to True-of-Date to ECI-J2000 was occurring in both sides of the receiver, thereby validating the receiver performance in all coordinate frames.

Characterization Test 3 - Spacecraft Slews

For this study, the same scenario was again employed however, both sides of the receiver were warm started, and a set of slews executed at every descending node. For each of four descending nodes a different slew profile was executed which characterizes all of the different slew profiles likely to be encountered by the EO-1 spacecraft. To allow sufficient recovery time after the last slew set, the run was extended to seven hours in duration instead of the usual six. The four slew sets executed were as follows:

Slew Set 1: At (DN-1)-22.5 minutes, execute a 6 min +90° Pitch (up).
At (DN-1)+12.5 minutes, execute a 6 min -90° Pitch (recovery).

Slew Set 2: At (DN-1)-22.5 minutes, execute a 6 min -90° Pitch (down).
At (DN-1)+12.5 minutes, execute a 6 min +90° Pitch (recovery).

Slew Set 3: At (DN-1)-22.5 minutes, execute a 6 min +90° Roll (right).
At (DN-1)+12.5 minutes, execute a 6 min -90° Roll (recovery).

Slew Set 4: At (DN-1)-22.5 minutes, execute a 6 min -90° Roll (left).
At (DN-1)+12.5 minutes, execute a 6 min +90° Roll (recovery).

While the receiver did not have time for the filter to fully converge before the first slew was executed, it became clear that the receiver in the flight configuration was quite capable of outputting valid navigation solutions during and after all the slew profiles even though there was no requirement for the receiver to be able to do so (in fact the baseline plan was to disable the use of GPS navigation data during the slews).

Characterization Tests 4 & 5 - Cold Start: Earth Point / Sun Point

The point of this pair of tests was to measure the time required to cold start the receiver in both the Earth pointing and Sun pointing attitudes. Limitations of the GPS Simulator prevented the execution of these final two tests before delivery of the flight unit to the EO-1 project. The GPS Simulator can simulate RF signals from only ten of the 32 GPS Space Vehicles (SVs) at any one time. The selection of which SVs the simulator provides is based upon the simulators knowledge of the users location and the assumption that the user antennas are aligned along the zenith direction. In the case of the canted orientations of the EO-1 antennas, this assumption is invalid. From a cold start, the receiver employs only one antenna at a time in a random search for nine of the GPS SVs. Since the antennas for EO-1 are not pointed in the zenith direction, the simulator is modeling SVs that the search antenna cannot see. Combine that with the random nature of the search employed by the receiver, and it becomes clear that the receiver may be looking for SVs that are not being modeled while many of the SVs being modeled cannot be seen. Any statements on the cold start performance of the receiver under these conditions would not be a fair assessment of the receivers capabilities for either the Earth pointing or Sun pointing cases.

Phase V: RPU/PCU Electrical Characterization:

The final phase of the testing involved confirming that the GPS receiver functions properly while obtaining its power from the Power Conditioning Unit (PCU) designed to provided proper power to the system while in flight. Up until now, all testing has been done with power provided to the GPS receiver from a GSE power supply. Since the spacecraft main power bus was deemed incompatible with the specifications of the GPS receiver, a power conditioner had to be designed to bridge the gap. This last set of tests were designed to ensure that not only does the PCU perform as expected, but that the GPS receiver performs in the presence of the PCU and to demonstrate the in-rush characteristics of the integrated GPS. The test setup is shown in **Figure 6**.

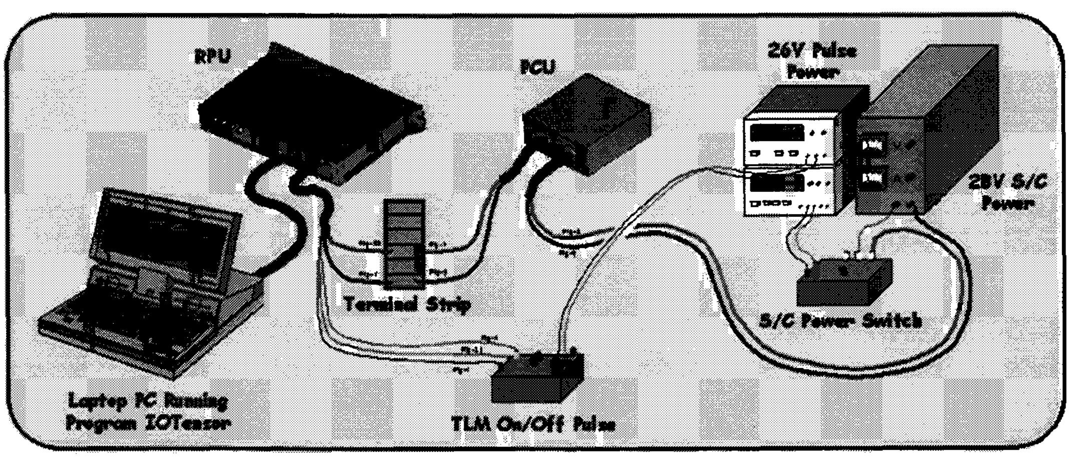

Figure 6: Test setup used to characterize in-rush and check PCU performance.

The simplest analogy to draw upon here is a simple desk lamp, which has a plug and a switch. To turn the light on, it is possible to turn the switch on and then plug the light in, or to plug the light in (with the switch off) and then turn the switch on. In the case of the GPS receiver, the connection to the 28 VDC main power bus is the 'plug', and the relay internal to the RPU is the 'switch'. Therefore, testing all the possibilities involves three separate tests:

1. RPU Relay = ON; S/C Power = OFF; S/C Power from OFF to ON (plug it in with the switch on).
2. RPU Relay = OFF; S/C Power = OFF; S/C Power from OFF to ON (plug it in with the switch off).
3. RPU Relay = OFF; S/C Power = ON; RPU Relay from OFF to ON (switch it on while plugged in).

Each of which were performed three times on each RPU. The first set was run on the RPU alone, so as to distinguish between characteristics of the RPU and those imposed by the presence of the PCU. The second set was run on the RPU/PCU combination with the probe located at the output of the PCU. The third set was run on the RPU/PCU combination with the probe located at the input of the PCU. During each test, a digital oscilloscope was employed to capture the curves which characterize the behavior of the system and were then included in a set of drawings which show the test set-up and resulting output of each test. The full test set was run first on the Engineering Test Unit (ETU) in combination with an early PCU breadboard many months before the delivery of either flight unit. These same tests were then performed on the RPU Side-A (the flight unit), and finally on the RPU Side-B (the flight spare) in combination with the flight PCU.

The ETU test set was run not only to test the PCU design (in breadboard), but to be sure the test was sufficient to fully characterize the flight unit, as well as to provide a set of baseline expectations. Comparison between tests yields a few notable differences between the ETU tests and the flight unit tests. For each Test-1, an oscillation is present on the plots of the flight unit that is not seen in the corresponding ETU test. This is due to the nature of the switch used to apply the 28 VDC spacecraft power. For the ETU tests, a solid state switch was used. This switch provided a clean start-up but introduced series resistance which would slightly lower the in-rush current. For the flight unit tests, a mercury switch was employed which has a noisier turn on characteristic, but would not limit the current as much as the solid state switch.

Other differences are best characterized by direct comparison of corresponding Test-3's which shows the relay applying power to the receiver. In the case of the ETU, a high frequency oscillation is present near the end of the power on cycle which is not present in either of the flight unit tests. This simply means that the relays in the flight unit are much cleaner (and therefore better) than those present in the ETU.

Conclusion of Phase V:

The in-rush tests performed fully characterized the power characteristics of the Receiver/PCU combination intended for flight on the EO-1 spacecraft. Up until this point, the two sides of the receiver were bolted to one another in a single fully redundant unit. At this time, the two sides of the receiver were separated from one another and a set of validation tests demonstrated that the separation was successful into dividing the formerly redundant dual receiver into two single string receivers dubbed side-A (the flight unit) and side-B (the flight spare). The GPS flight hardware (side-A) was then delivered to the EO-1 project on June 22, 1998 and subsequently integrated into the spacecraft. Once integration of the GPS hardware was complete, the three Phase II tests were repeated this time using the spacecraft power as a power source to the PCU and the GPS receiver. This repeat proved that the Receiver/PCU combination would operate as expected and that the GPS hardware had been successfully integrated into the EO-1 spacecraft.

Since all simulator testing to date employed the same scenario developed earlier, it was mutually agreed by all involved that checking the system against another epoch would be a wise precaution and so another scenario was developed. This new scenario was run using the flight spare receiver (so-called Tensor B) before the simulator was moved over to the spacecraft staging area. In this way, not only could the scenario be checked, but the performance of the flight receiver anticipated. This scenario would accomplish several goals in a single stoke: it would allow the receiver to demonstrate its performance at a time other than the original test time, it would verify receiver performance in an epoch after the GPS system rollover (which occurred the morning of August 22, 1999), it would demonstrate the ability of the receiver to fly through the transition to the year 2000 (the so-called Y2K issue) and finally, it would provide the ground its first opportunity to command, configure and warm start the receiver without help from any GSE whatsoever.

Having completed these tests, if was determined that the integrated EO-1 GPS was functioning in a manner consistent with the performance seen prior to delivery, that is to say the integrated EO-1 flight GPS system was functioning properly and the ground system had demonstrated its ability to correctly interface with the GPS to command and configure the system as well as receive and employ navigation from the system. The EO-1 GPS was now ready for spacecraft level testing in the thermal vacuum chamber.

Thermal Vacuum Test 1:

During the first run at thermal vacuum testing (November 1999), the GPS was run for prolonged periods (+24 hours) of time at both hot and cold plateaus. The system performed well, when it was provided sufficient signal from the GSE setup used to relay signals from the roof of the building into the test chamber. Several dropouts were seen and a problem report generated.

GSE Upgrade:

As a result of unrelated piece part problems with the EO-1 spacecraft, thermal vacuum testing would need to be run again. It was decided that this time a lesson would be learned from the first set of tests and the RF setup used to carry GPS constellation signals from the roof to the test chamber would be modified so as to minimize the possibility of interruptions or dropouts. The GSE setup used during TV-1 employed a rooftop GPS antenna followed by a 3 meter run of RG-142 coaxial cable. This led to a pair of power amplifiers whose output was sent to the test chamber via approximately 200 ft of Belden 9913 low noise cable. The end of this cable was then connected to one of four inputs to the test chamber. Four 6 meter runs of RG-142 cable connected the inputs to the chamber to the flight antennas via four hat couplers spaced approximately 6" from the flight antenna.

RF experts examined the TV-1 setup and found plenty of room for improvement. It was discovered that the signal losses associated with the RG-142 cable were significant over long distances, so it would be eliminated wherever possible. This was done by constructing a weather resistant box which housed the rooftop antenna externally and contained a low noise amplifier (Down East Microwave model 520LNA20WP) and a gain amplifier (Mini-Circuits model ZEL-1217LN), thus eliminating the 3 meter run of RG-142 originally used to keep the powered amplifiers from being exposed to the weather. This cable it was found, reduced the input signal by 3dB before it ever got to the first amplifier, so it's elimination was crucial to improved performance. The 200 ft run of Belden 9913 low noise cable was retained since 10dB lost in the run was more than made up for by the 40dB of combined amplifier gain provided by the previous change. Although they could not be eliminated altogether, the RG-142 cables in the test chamber itself were minimized to 3 meter runs. Finally, the gaps between the GSE antennas and flight antennas maintained by the four hat couplers was reduced to less than 0.5". The modified setup was then checked against the flight spare receiver before connecting it to the flight system. The net result of these changes were strong clear signal to the GPS receiver which allowed for consistently quick warm starts (~10 mins) and continuous lock on the GPS constellation. The setup was declared both safe and functional and was then connected to the flight system where behavior identical to that of the backup receiver was noted.

Thermal Vacuum 2:

For TV-2 four data sets would be collected. The first set (June 28, 2000) would baseline the behavior of the GPS connected to the new GSE setup while the spacecraft was outside the test chamber. The receiver was warm started and signal strength data was collected using a laptop PC receiving telemetry directly from the receiver via the EO-1 skin connector. A minimum of 10 minutes of data was collected along each of the four antenna paths into the flight receiver. The constellation provided uninterrupted navigation signals the entire time by maintaining constant lock through all four paths. Needing signals from only four GPS satellites at any time to provide a navigation estimate, the flight GPS maintained constant lock on no less than six (most commonly seven) throughout the test.

The second data set (July 7, 2000) was collected during a cold plateau phase of TV-2. Because the time was available, an hour's worth of data was collected through each antenna path even though a minimum of 10 minutes is all that is required. During this test as well, strong signal strengths from no less than six (most commonly seven or eight) GPS satellites were observed with no signal dropouts or interruptions to navigation.

The third data set (July 10, 2000) was collected during a hot plateau phase of TV-2. This time, there was insufficient time for the luxury of collecting a full hour from each antenna path, so an hour was collected from antennas 1 & 3 and 30 minutes from antennas 2 & 4. Here again, strong signal strengths from no less than seven (most commonly eight or nine – the maximum the receiver can process) GPS SVs were observed with no signal dropouts or interruptions to navigation.

Finally, the last data set was collected (July 21, 2000) after the spacecraft was removed from the thermal vacuum chamber. A minimum of 10 minutes worth of data was collected from each of the four antennas to replicate the pre-Thermal Vacuum test data and verify that there were no observable changes in the behavior of the system as a result of the thermal vacuum tests. The success of this test meant that the EO-1 GPS pre-launch testing was complete and ready for the spacecraft to be shipped to the Launch facility for the expected launch on October 17, 2000.

EO-1 LAUNCH & EARLY OPERATIONS

The EO-1 spacecraft was launched with its companion payload, SAC-C, aboard a Delta II 7320 ELV on Tuesday, November 21, 2000 at 18:24:25.083 (UTC) from Vandenberg Air Force Base. The launch was very close to nominal with EO-1 separated into its insertion orbit at 3600 seconds MET (Mission Elapse Time) and SAC-C separated at 5444 seconds MET. At the time of the launch, the GPS receiver was powered off.

During orbit three, at approximately 6:15 pm local time (23:15 UTC) while the spacecraft was still in the Sun pointing mode, the GPS receiver was powered up and configured for flight operations. As expected, the receiver provided no information during the first 30 seconds while it went through its bootstrapping routines. Once complete, telemetry indicated that the receiver was powered on, functional and searching for SVs. Again, as expected, the receiver found its first GPS SV at 6:21 pm (23:21 UTC) at which time collection of the GPS almanac began. By 6:25 pm (23:25 UTC) the receiver had found a second GPS SV increasing the probability that complete almanac collection would be successful the first time. At approximately 6:33 pm (23:33 UTC) the "ALMANAC OK" flag was set in telemetry indicating that the almanac collection process had been successfully completed. Based on simulations run prior to launch, it was not at all certain that the receiver would immediately acquire any more GPS SVs or whether an ephemeris would have to be uplinked. Just as this point was being made, the receiver telemetry indicated that a third GPS SV had been found and finally a fourth. The GPS receiver successfully cold started and began providing navigation solutions at approximately 6:38 pm (23:38 UTC) demonstrating that not only could the EO-1 GPS receiver execute a cold start, but it could accomplish it inside 20 minutes while still in the Sun pointing mode. This was better performance than anyone (including the vendor) expected. The expectation was low since the EO-1 GPS antennas are pointed in a non nominal off zenith orientation to avoid interference from the payload attach fitting. As expected, SV dropouts occurred during spacecraft night as the orbit carried the Sun pointed spacecraft through an orientation where the GPS antennas were pointed primarily at the Earth.

The following day, a test of the Safehold system was executed and the GPS receiver was automatically powered down as part of Safehold load shedding. Once the test was successfully completed, the receiver was again powered up (while still in the Sun pointing mode) and again was able to cold start inside 20 minutes. The implications for the future are significant in that no uploads would be necessary to initialize ephemeris processing on EO-1 during any future safehold recoveries. The receiver remained on and provided navigation fixes throughout the next several days which included all of the EO-1 gyro calibration slews, Delta-V slews, and imaging slews.

By Friday morning operations engineers reported that the receiver errors (as measured against ground measurements) were consistently inside 100 meters (see **Figure 7** where the residual values are in km) so that the ACS was ready to use the GPS in the active control loop. At approximately 1:19 pm local time (18:19 UTC) on Friday November 24, 2000, EO-1 became the first spacecraft operated by GSFC to employ GPS data directly as the main source of ephemeris data for on board navigation and attitude control.

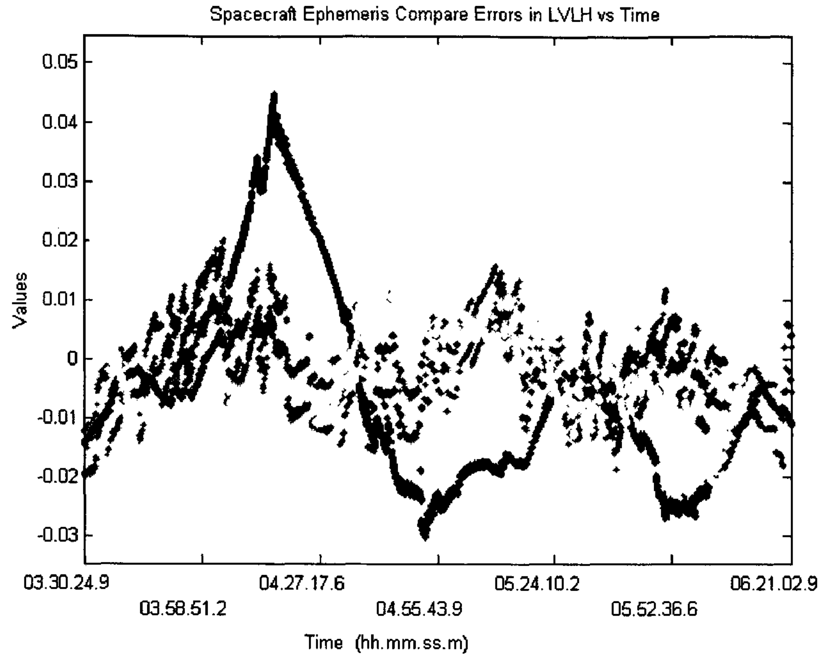

Figure 7: On Orbit EO-1 GPS Residuals (km).

The First Anomaly:

During the weekend of December 16 & 17 the receiver began losing lock on the GPS constellation frequently enough to cause the spacecraft to remove the GPS from the active control loop and begin to use ephemeris propagations for navigation. Examination of GPS data demonstrated during the times in question, there were periods when the number of GPS SVs being tracked dropped below the minimum requirement of four and even as low as zero. A more detailed investigation revealed that the watchdog timer in the GPS receiver itself reset late on the evening of Saturday Dec 16. When this watchdog timer reset, it presumably reset the receiver configuration to its default values rather than those employed by EO-1. The designed function of the Tensor™ is to provide both navigation and attitude data. To accommodate this, the default configuration of the receiver outputs navigation data at a 1 Hz rate and attitude data at 10 Hz. Since EO-1 is not using GPS attitude data (the non-nominal antenna pointing will not permit it), the receiver defaults to outputting zeros in the attitude packet at 10 Hz. To cut down on the transfer of data useless to EO-1, this rate is reconfigured to its lowest rate of 1 Hz (it cannot be set to 0 Hz). When the watchdog timer reset, the attitude packet update rate went back up to the default 10 Hz rate effectively 'clogging' the data bus causing the spacecraft to not receive navigation updates at the expected rate which, in turn, caused the spacecraft to question the GPS output and revert to ephemeris processing. By Monday, it had become clear that the watchdog timer had reset, and so it was decided to reset and reconfigure the receiver itself. Once this was done, all anomalous behavior subsided and the GPS was again running as the main source of navigation data in the active control loop.

Later investigation revealed a curious cause for the problem. The GPS receiver computes time by counting the number of milliseconds during each week. At the beginning of each GPS week, this counter rolls over as the GPS week number increases by one. Internal to the receiver is a Kalman filter which filters the navigation data , retaining a diminishing memory of the navigation data from previous iterations. This is what allows the receiver to compute navigation solutions during those periods where the number of visible GPS SVs drops below 4. When the millisecond clock counter rolls over, the Kalman filter attempts to process over what it perceives to be a very large time interval, violating a 10 second firmware limit, tripping in turn the watchdog timer. The result is, effectively, a reset of the receiver with a bit being set that lets operators know it was the watchdog timer that called for the reset. The timing is such that this kind of reset can occur every 10 – 20 weeks. The vendor recommended operational fix was to disable the Kalman filter for a short time during the weekend rollover. As the problem has not recurred on the EO-1 the

present recovery plan (should this happen again) is to simply reset the receiver and re-enable GPS processing one orbit later. This is seen as of minimal impact to the EO-1 mission, since as an experimental spacecraft it is not constantly performing science observations.

The Second Anomaly:

Shortly after the watchdog timer problem was solved, another curious problem arose. This time, the spacecraft removed the GPS from active control due to what it reported as repeated "GPS packet validity failures". What was curious about this failure was that it occurred at the same time each week – which seemed to point to another issue related to the GPS millisecond clock counter. Investigation here revealed the problem to be not an issue with the receiver, but with how the data from the receiver was being interpreted.

The Tensor™ receiver is capable of outputting many data packets which are flexible in length. Distinguishing between packets therefore is not a simple case of counting the bytes in a given packet. Instead, the receiver is programmed to key on the data value 16 in the data (in hexadecimal 'x10', the x indicating the value is in hex). When the hex value 'x10' is encountered alone, it is assumed to indicate the beginning of a data packet. If the value 'x10' is intended as a legitimate data value, the receiver software places a second 'x10' in the data stream after the first in a process called 'DLE stuffing'. If the hex value 'x10' appears in the hex byte 'x1003' it is assumed to indicate the end of a packet. The spacecraft housekeeping RSN looks for these end-of-packet indicators to know where to segregate the data stream into its constituent packets. While the Tensor™ includes the DLE stuffing in all of the housekeeping telemetry packets, the navigation packet being of a fixed length does not include DLE stuffing. Since the spacecraft RSN software was written under the assumption that all packets include DLE stuffing, whenever the hex sequence 'x1003' is encountered in the navigation packet without DLE stuffing, the housekeeping RSN assumes the receiver is signaling the end of a packet and divides the data stream accordingly. The incorrectly divided packet fails the validity check and the data contained in the packet is ignored. If this happens in another area of the navigation packet, a single packet is lost, there is no impact to the spacecraft. The last valid update is propagated and the next valid output of the GPS receiver is used. If too many of these occur simultaneously, as is the case when the upper bytes of the millisecond clock counter read 'x1003', a 60 second timer limit is violated and the housekeeping RSN informs the spacecraft to ignore subsequent output from the GPS receiver.

Once a week the millisecond clock counter indicates that the current time is 3 days 2 hours 37minutes 12.064 seconds into the current GPS week. At that time, the number of milliseconds into the GPS week is 268632064 or in hex 'x10030000'. As can be seen, the hex byte, 'x1003' appears in the MSB of the clock value and is present in the clock for a total of 65.536 seconds. The housekeeping RSN interprets the 'x1003' in the clock word to mean the end-of-packet has been reached. It then breaks the valid packet in two invalid packets for the next 65.536 seconds until the hex value of the clock word reaches 'x10040000'. Since the duration of the problem violates the 60 second data valid timer, the spacecraft removes GPS from the control loop and enters the ephemeris propagation mode.

The best technical solution would have corrected the housekeeping RSN software to distinguish between those packets that employ DLE stuffing and those that do not. However, at this point with the spacecraft flying and the software otherwise working perfectly, this was seen as too risky, time consuming and expensive, and so easier solution was sought. It was known that the problem persists for no more that the 65.536 seconds once per week so changing the valid data test timer from 60 seconds to 120 seconds would bypass the problem. Although 65.536 seconds of navigation data would be ignored once per week, the propagator used by the spacecraft during this time would not accumulate enough error over that minute to violate any mission constraint. Once the timer duration was reset, there were no subsequent occurrences of this problem.

EO-1 GPS Performance Versus Ground Based Ephemeris

Ground based definitive orbit determination (OD) solutions are generated for EO-1 every Monday, Wednesday and Friday using S-band Two-Way Doppler tracking data from ground stations at Wallops Island, Virginia; Poker Flats, Alaska; and Svalbard, Norway. Approximately ten EO-1 S-band passes are taken per day with each pass consisting of about ten minutes of valid Two-Way Doppler data (above 10 degree elevation). Ground based OD solutions are generated with STK/PODS using a 70 by 70 JGM-2 Earth Model, Sun/Moon perturbations, and applying a solar radiation pressure (Cp) of 1.5 and solved for coefficient of drag (Cd). In addition to solving for position, velocity and Cd at epoch in the ground OD process, ground station Doppler biases are solved for as well.

Once the process is complete, the final OD solution state is read into AUTOCON-Gto generate the ground based ephemeris, which contains definitive and predictive ephemeris information. GPS position data is extracted from the EO-1 telemetry and then compared with the definitive ephemeris to determine the performance of the on-board GPS ephemeris versus the ground based ephemeris. Typical differences are shown in **Figure 8.**

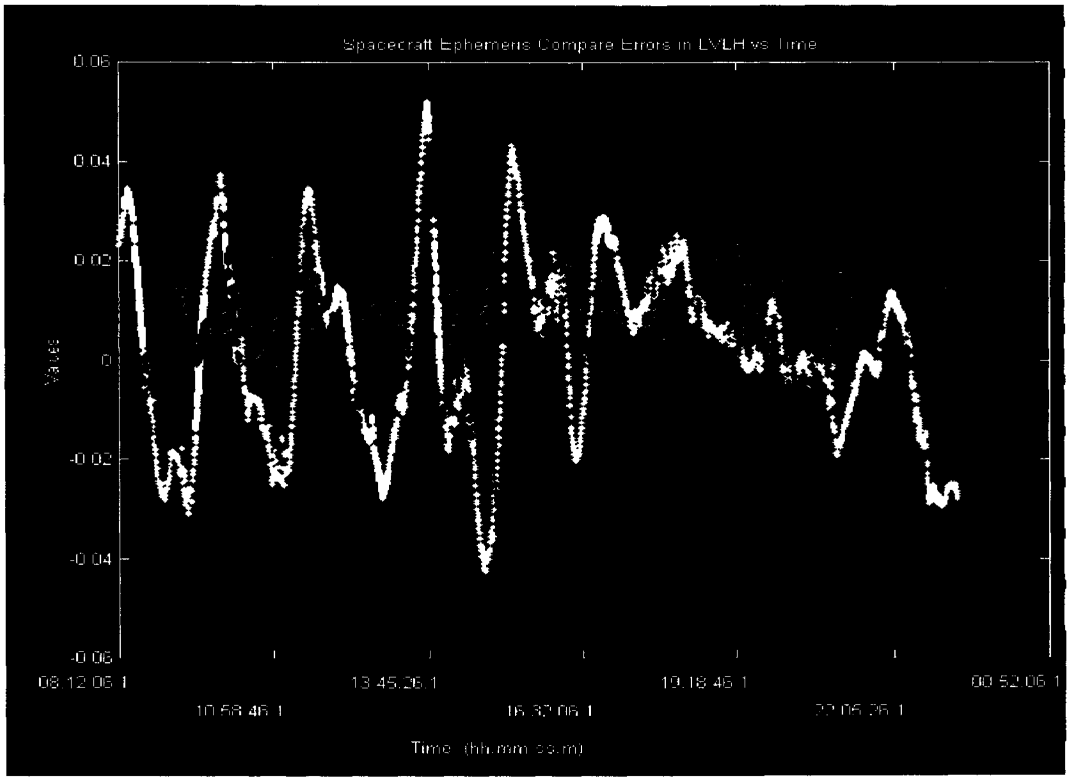

Figure 8: Position Differences Between GPS and Ground OD Over 16 Hours.

Position Component	Mean Difference	Standard Deviation
Along Track (meters)	6.379	15.279
Cross Track (meters)	4.625	18.111
Radial (meters)	0.171	10.934

Table 5: GPS/Ground OD Difference Statistics.

GPS performance degradations over 100 meters can be observed during attitude maneuvers that take EO-1 significantly off nadir. Off nadir pointing means that the GPS antennas may not have a clear line of sight to all the available SVs and navigation performance suffers as a result. Occasionally the EO-1 spacecraft is oriented so as to make science observations of the Sun and Moon. This is done primarily to calibrate the spectral output of the science instruments against the known spectral characteristics of these bodies. For this reason, the orientation maneuvers that direct the science instruments to these targets are referred to Solar and Lunar calibration slews. The following **Figures 9 & 10** show the effects this non-nadir pointing of the EO-1 spacecraft has on GPS performance. The data was compiled over times when EO-1 was performing a Solar Calibration Slew and a Lunar Calibration Slew, respectively.

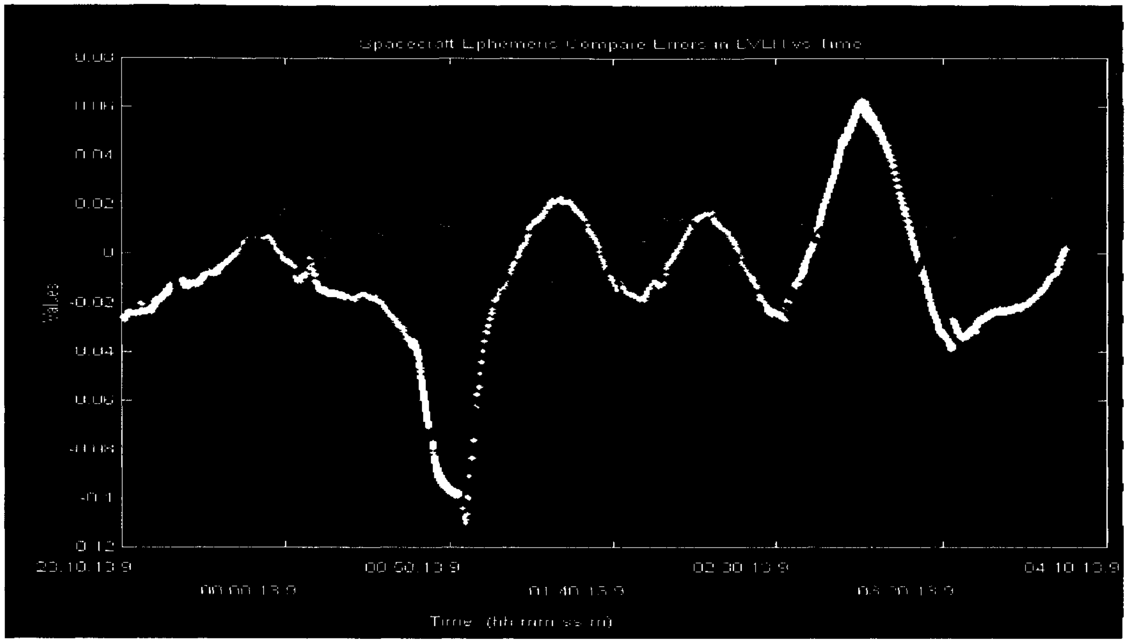

Figure 9: GPS / Ground OD Differences During Solar Calibration Slew.

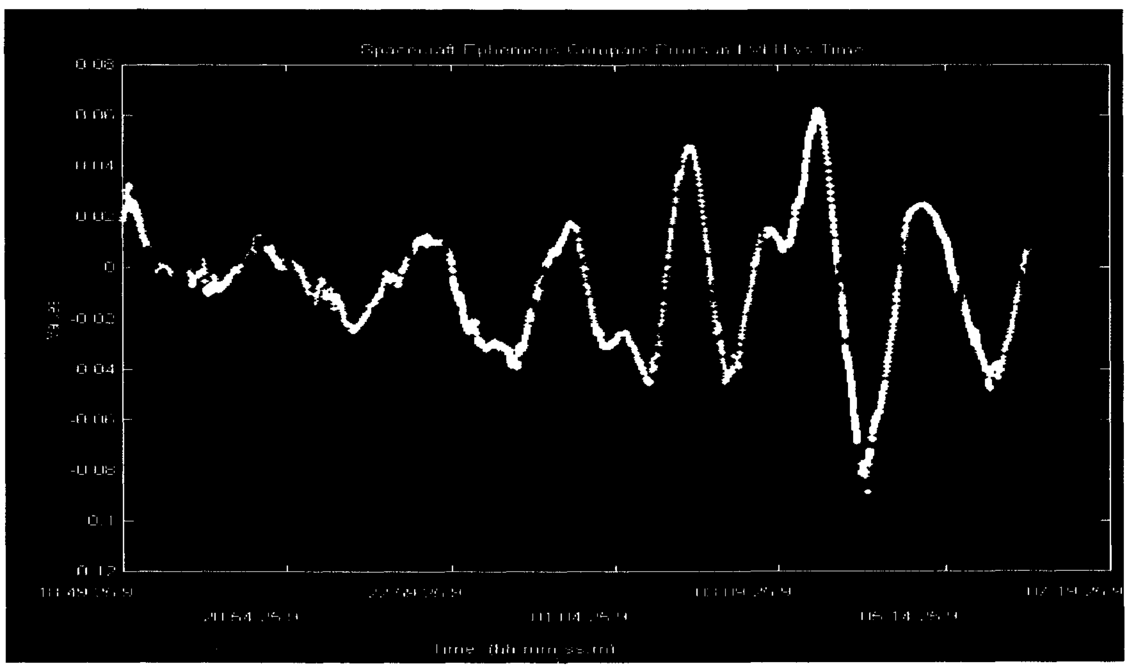

Figure 10: GPS / Ground OD Differences During Lunar Calibration Slew.

CONCLUSIONS

The EO-1 experience is proving GPS to be a worthy addition to the battery of spacecraft subsystems available for future use. It must not be forgotten however that GPS navigation in space is still a relatively new process and will require a learning curve before the full potential of its capability can be exploited. As with any new system, special care must be taken to fully understand the specific receiver being used and how to employ its output properly. For the future, a close relationship between the manufacturer and the user can serve to close the loop and assure future generations of users gain the benefit of past user experience. Although the EO-1 experience turned out to be quite successful, specific lessons can be learned which can improve the process for the next generation of spacecraft designers and builders.

- GPS relies quite heavily on RF communications. It is therefore important to bring in experts in RF communications from the beginning, especially when designing the environment in which the validation testing is to occur.
- GPS relies quite heavily on very accurate timing. To take full advantage of the atomic clock accuracy available by using GPS, the testing facilities should have equipment capable of measuring GPS signals to an accuracy beyond that of the GPS itself.
- To fully simulate the expected performance, GPS Simulators should be capable of modeling the entire sky as seen by the user spacecraft. This would require a GPS simulator capable of providing simulated signals from as many as 15 or more satellites of the GPS constellation.
- Because the software in any GPS receiver is as complex as it is, a close working relationship between the manufacturer of the GPS receiver and the ultimate user is essential to assure that the user gets the best performance that the GPS receiver can deliver.
- Maintaining a constant level of GPS performance requires that either the spacecraft always be oriented for optimal visualization of GPS SVs or the GPS be augmented with additional antenna capability to accommodate non-nominal attitudes.

As time moves on, spacecraft are becoming increasingly numerous and complex. Keeping costs down under these circumstances inevitably calls for increasingly autonomous spacecraft. This mindset makes it a virtual certainty that the list of spacecraft looking to employ GPS will continue to grow almost as quickly as missions manifest themselves. With the continued success of EO-1 and other spacecraft using GPS as the primary means of navigation, it can be expected that GPS will finally take its place in the arsenal of 'tried and true' technologies for the future of satellite design.

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